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Materials for Space-Formed, Rigidized Structures Used in Solar Thermal Propulsion Concepts[1]

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Large, inflatable, solar collector support structures that can be conveniently stowed, deployed, and efficiently rigidized are important components of solar thermal propulsion systems. Spacecraft powered by solar propulsion engines will be able to provide the velocity change required to economically maneuver large payloads from one orbit to another. This is an innovative concept, that when applied, will double the efficiency of currently used LH_2 / LO_2 chemical engines.

The Air Force Research Lab (AFRL) has funded Thiokol Propulsion to develop rigidized solar collector support structures. This paper will address the properties of materials used for the in situ construction of rigidized structures in space. Photo-polymerization, thermal curing using resistive heating and the use of thermoplastics will be discussed. Space processing of polymeric materials presents significant issues including outgassing, required cure energy, and the thermal extremes of the orbital environment. These issues will be addressed and the current status will be reviewed.

INTRODUCTION

Using the sun's energy to propel a spacecraft requires large solar energy collectors. Figure 1 shows a conceptual view of a solar thermal rocket featuring inflatable solar concentrators supported by inflated and rigidized struts.

Solar thermal propulsion (STP), a concept that was

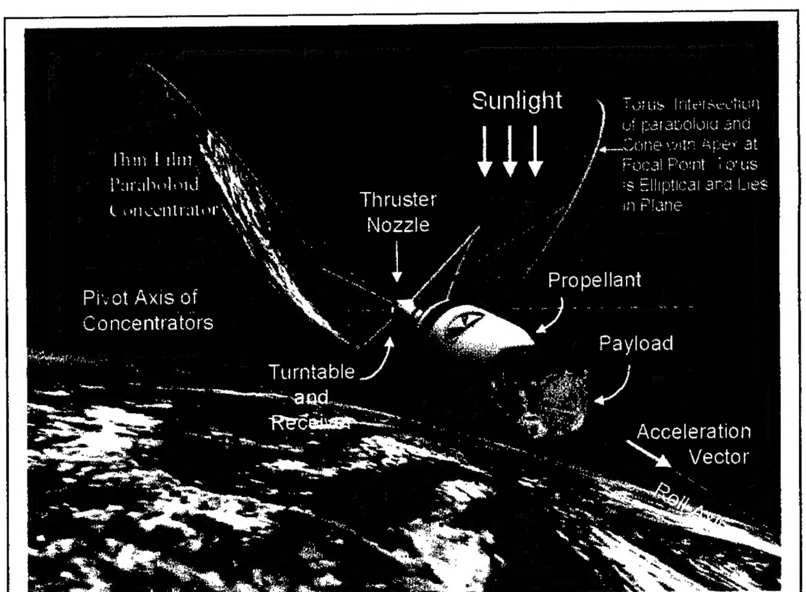

Figure 1. Solar Thermal Propulsion Rocket AFRL

introduced by Erlick in the 1950's[1], requires the sun's energy to be transferred to a low molecular weight fuel such as hydrogen. The thermal energy stored in the hot fuel is then converted to kinetic energy by expansion through a diverging nozzle. This results in a high efficiency (800 – 1,000 sec Isp), low thrust (2-10 lb force) propulsion system[2]. Spacecraft powered using STP systems have been proposed for orbital transfer, interplanetary, and other delta velocity missions.[3]

The use of large lightweight solar collectors that can precisely focus the sun's energy on the engine in the space environment presents many design and materials challenges. The harsh orbital environment with very low pressure, radiation, ram atomic oxygen (AO), ion, and micrometeoroid impacts significantly degrades most polymeric materials.

This paper will focus on a discussion of materials used for the in situ construction of rigidized structures in space. Rigidization methods examined include photo-polymerization, resistive heating, thermal cure, and thermoplastics.

TECHNICAL DISCUSSION

The Solar Thermal Propulsion Critical Flight Demonstration (STPCFD) program is an Integrated High Payoff Rocket Propulsion Technology (IHPRPT) demonstration program. The objective of the program is to demonstrate through earth based testing that an inflatable concentrator-based STP system is ready for a space experiment demonstration.

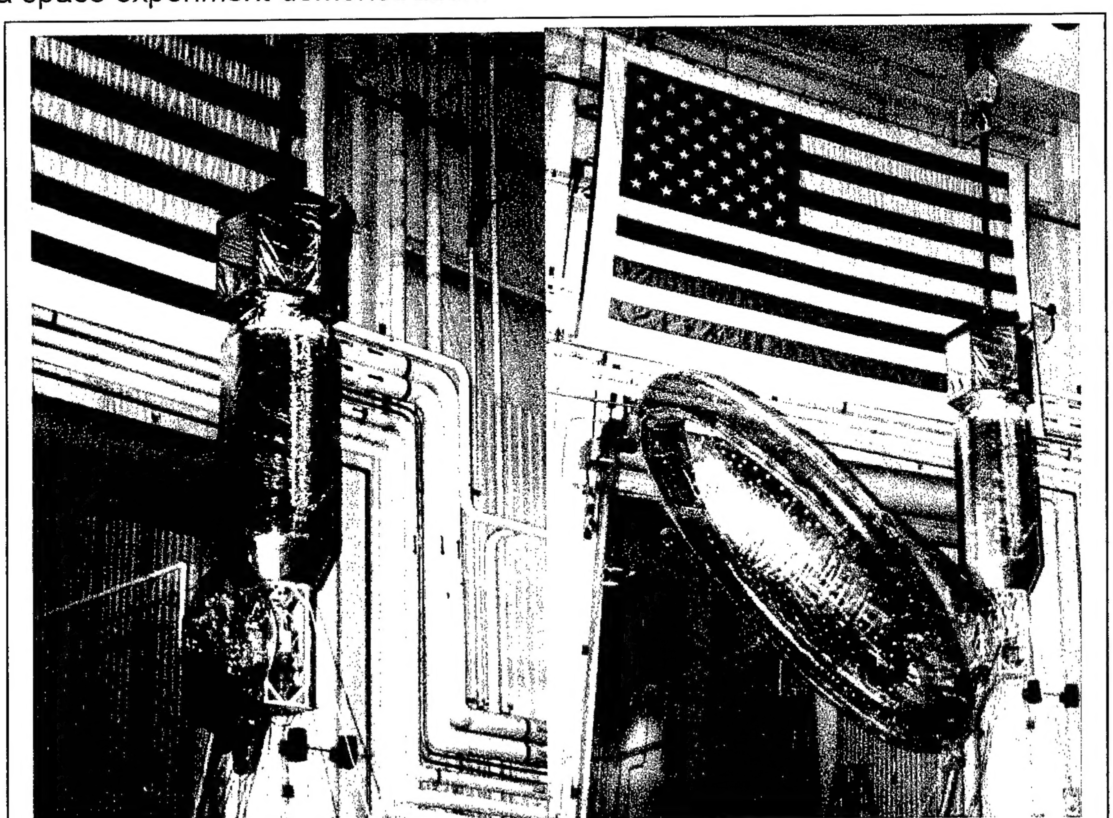

Figure 2. Pre and Post-deployed 2 X 3 Meter Inflatable Concentrator Assembly

Large, inflatable, solar collector support structures that can be conveniently stowed, deployed, and efficiently rigidized are important components of an inflatable, solar thermal propulsion system. Recently, a 1/3 scale space experiment prototype underwent deployment testing in Thiokol Propulsion's manufacturing facility near Brigham City, Utah. Figure 2 shows the inflatable concentrator in the pre-deployed-packaged and post-deployed configurations.

Inflatable/rigidizable struts were used to connect the torus and concentrator to the spacecraft interface ring. Line drawings of this apparatus are shown in Figure 3. The strut structure will be the focus of this paper. They are

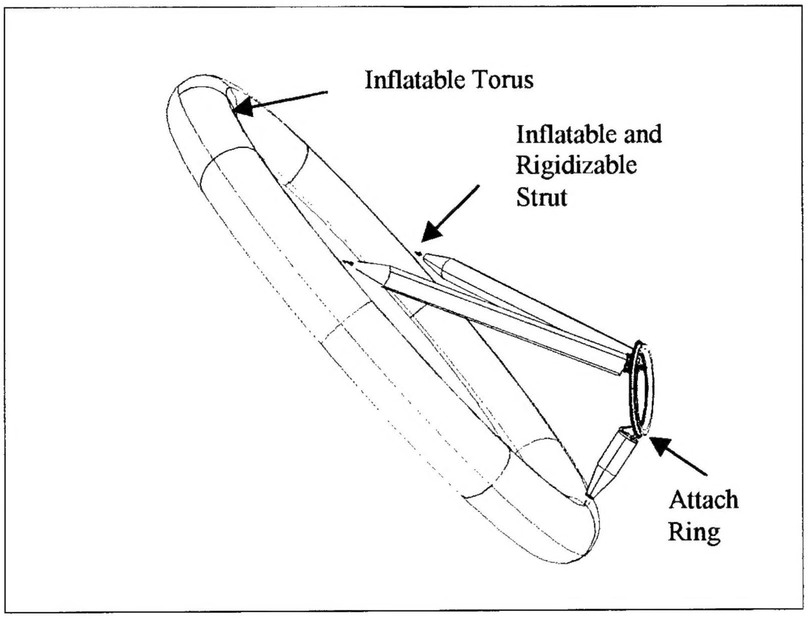

Figure 3. Line Drawing of Inflatable Concentrator Support Structure

composed of a resin-impregnated composite fabric layer sandwiched between thin film polyimide (CP-1) skins. CP-1 is a space-qualified material[4] produced by SRS Technologies. The inner thin film is pressurized forming the male mold for the rigidized structure. The outside film functions as the female mold. The pre-impregnated composite fabric is the filling of the sandwich and becomes compliant to the mold shape during inflation.

Space Processing Of Polymeric Materials

Processing polymeric materials in Low Earth Orbit (LEO) presents challenges associated with the environment. The thermal environment, low pressure, microgravity, and particle collisions are of primary concern. For the purposes of this paper the thermal environment and, to a lesser extent, low pressure issues will be addressed. The dynamics of deployment in microgravity

and ablation from particle collision are important topics[5], however, they will not be addressed.

Thermal Environment. A test was conducted to determine the thermal response of a strut in a space-simulated environment. Strut segments 5.5 inches in diameter and 22 inches long were prepared for thermal testing. These beam segments were instrumented with 6 thermocouples. One thermocouple (TC) was placed in each of the four circumferential quadrants. Two additional thermocouples were placed in line with the thermocouple in the quadrant directly under lamps. All thermocouples were sandwiched between laminate layers of the composite. Both heavy fiberglass (~0.066 inch wall thickness) and light fiberglass (~0.030 inch wall thickness) strut sections were tested. The tests were performed inside a vacuum chamber equipped with a liquid nitrogen cold wall, and sun lamps. The beam was placed in a position between the cold wall and the lamps so that it was exposed to a flux of ~1,320 W/M^2. It should be noted that the chamber wall adjacent to the 270° TC is cooler than the opposite wall because the liquid nitrogen lines that feed through it. This accounts for the delta in the TC readings at the 90° and 270° location.

The chamber pump was started and the pressure was allowed to stabilize at approximately 100 millitorr. The cold wall was then activated and the temperature was allowed to stabilize. The sun lamps were turned on and the thermocouple temperatures were recorded for approximately three hours. A near steady-state thermal response of the beam was achieved within the first 15 minutes. Figure 4 shows the response of the heavy weight glass beam. Figure 5 shows the response of the lightweight glass beam.

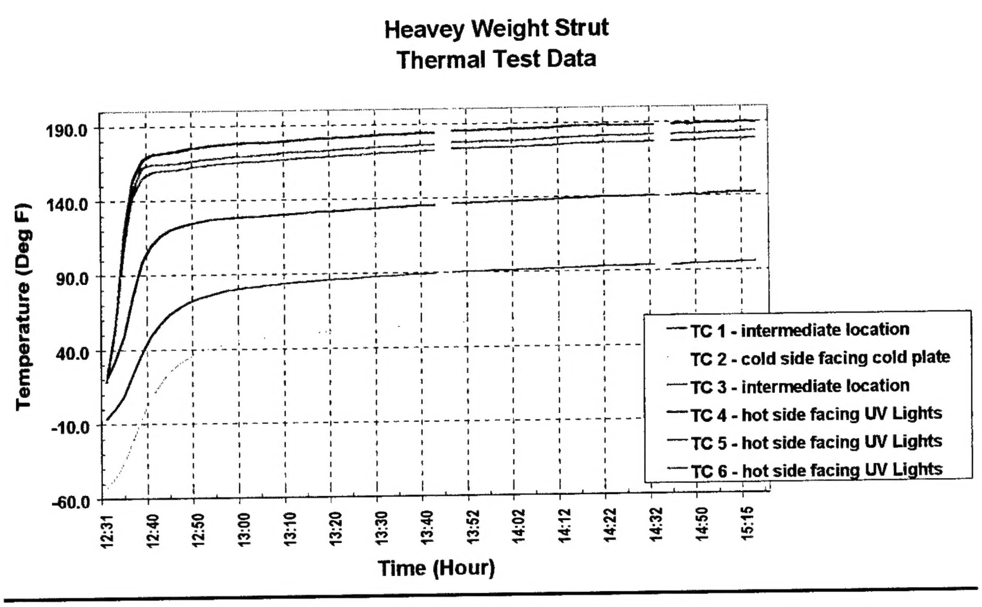

Figure 4. Fiberglass Beam (0.066 inch wall) Response to 1,320 W/M^2
(TC-4, TC-5, and TC-6 were placed at 0° and inline with the UV lights TC-1 was placed at approx. 65°, TC-2 was at approx. 150°, and TC-3 was at approx. 260°)

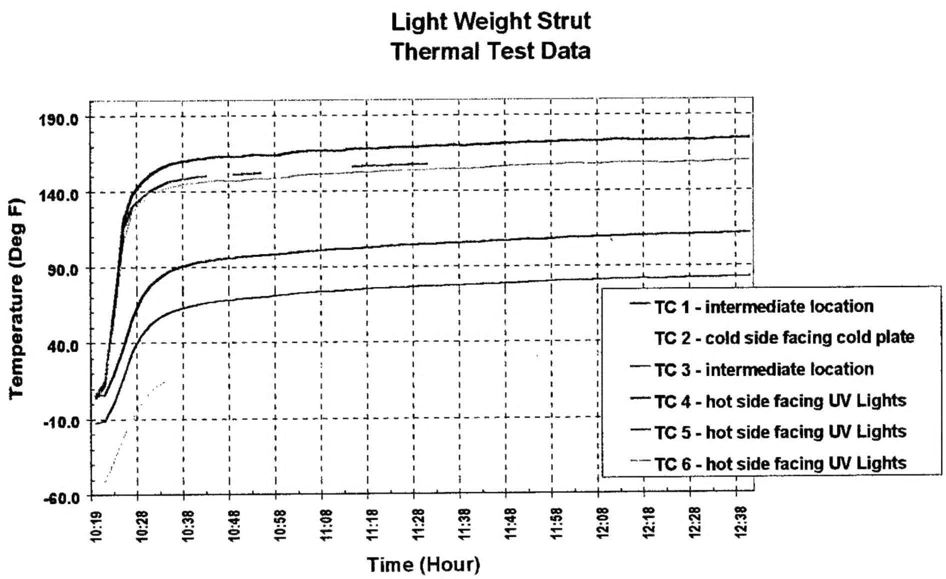

Figure 5. Fiberglass Beam (0.030 inch wall) Response to 1,320 W/M^2
(TC-4, TC-5, and TC-6 were placed at 0° and inline with the UV lights TC-1 was placed at approx. 90°, TC-2 was at approx. 180°, and TC-3 was at approx. 270°)

Thermal Modeling. Ideas Master Series Thermal Model Generator (TMG) was used to generate a radiation thermal analysis of the beam tests. These results indicate that the strut material and configuration can be effectively modeled. The results, shown in Figure 6, are presented in the form of an isotherm plot for the strut in steady-state conditions. Also plotted is the thermal response for the hot and cold sides for measured versus calculated temperature (Figure 7). A solar (UV) absorptivity of 0.9 and transmittance of 0.1 were used in the calculations. These values correlate with experimentally determined measurements on the fiberglass strut material.

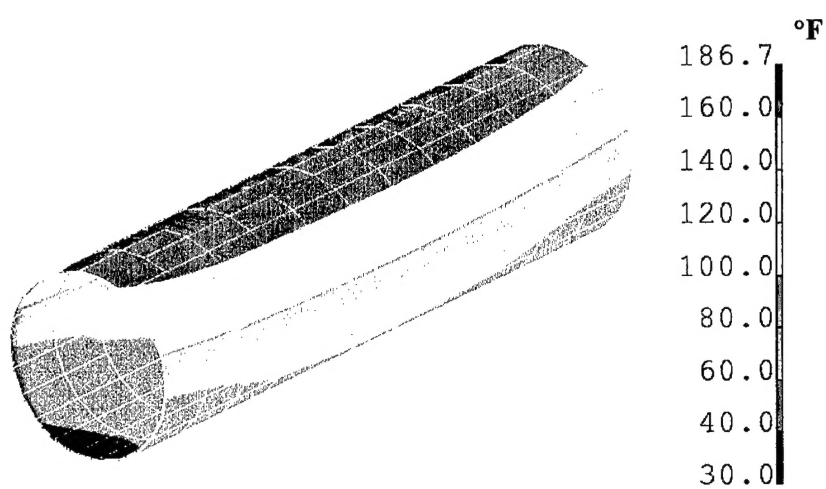

Figure 6. Model Thermal Response as a Steady-State Isotherm Plot

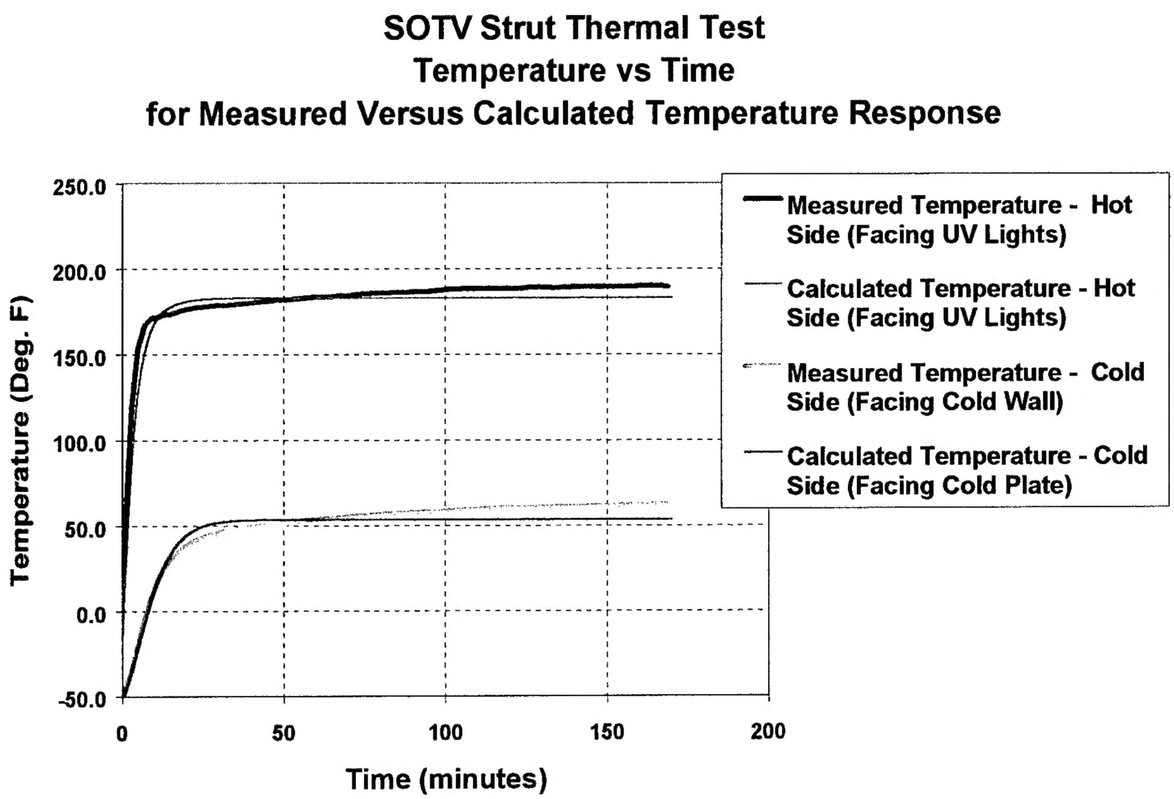

Figure 7. Measured vs Calculated Thermal Response

Materials. T-300 graphite and fiberglass fabrics have been evaluated for use in construction of inflatable/rigidizable struts. Several resin systems have also been evaluated for the composite matrix material. These include: three ultraviolet curing resins Sartomer's CN -104C 75, CN -120 S80 and CN - 983, a thermoplastic (polystyrene), and a thermoset (TCR UF3325). The first two UV curable resins are epoxy-based and the third is a urethane. All of the UV resins have acrylate functionality and are UV-curable.

Ultraviolet-cured materials, as the name implies, are cured using ultraviolet light thru the mechanism of free radical polymerization. A UV curable composite structure is first deployed by inflating it to achieve its functional shape and then rigidized by exposure to UV light. Glass cloth allows UV light to penetrate the composite resulting in complete curing. This structure may be protected from unwanted solar heating with a covering of multi-layered insulation (MLI) installed after curing. Materials like graphite cloth are opaque to UV light, and prevent the cure of shaded resin. This limits the use of UV curing to structures that can be formed from UV transparent or partially transparent materials.

Thermoplastic materials can be repeatably softened and hardened by heating and cooling the temperature about their softening points because they

undergo physical rather than chemical changes when heated.[6] The deployment and rigidization of a thermoplastic composite structure involves heating the packaged structure above the softening temperature of the thermoplastic resin matrix. Once the resin is softened, the structure can be deployed by inflation. Rigidization is accomplished by removing the heat source and allowing the thermoplastic matrix to cool below its softening point. This structure can be protected from solar heating with a covering of multi-layered insulation (MLI).

The solar flux of approximately 1,350 W/M^2 in LEO brings composite beam materials to a temperature of 190°F rather quickly. This temperature is sufficient to soften some thermoplastics. With electric resistive or pyrotechnic thermal augmentation the melt temperatures of most thermoplastics can be reached. This is a reasonable approach, however, the mechanism for shielding the composite material from the sun after the softening is accomplished adds significant complexity to the system.

A less complex system involves covering the thermoplastic composite skin in MLI in the fabrication process. Softening of the thermoplastic for deployment is accomplished by applying electric current directly to carbon fibers of the structure fabric, and/or through heating insulated, resistive wire. Laboratory evaluation of this concept has shown that 0.74 watts per square inch is required to raise the temperature of two layers (0.018 inch per layer) T-300 UF3325 TRC™ prepreg 200° F through resistive heating of the carbon fiber. Whereas, 2.9 watts per square inch is required to raise this same carbon material 200° F through insulated resistive wire. It should be noted that these tests were performed on a laboratory bench and the samples were not insulated. Much of the heat generated was lost to the ambient environment through convection and radiation. A temperature rise of 200° F is required for a period of 10 minutes for deployment of the structure. Typically two long beams, (3 inch diameter 18 foot long), and one short beam (3 inch diameter 4 foot long) are used to support a solar concentrator. The power demand to soften the structure for deployment is too great for the start-up batteries of existing spacecraft.

Thermoset materials are cured by application of heat. The applied energy accelerates the chain extension and/or cross linking reactions resulting in rigidization. Typical epoxy or polyester resin composites are examples. Thermoset materials typically have service temperatures at or below the cure temperature.

The deployment and rigidization of a thermoset composite structure will involve deploying the uncured structure into its functional shape. Rigidization follows the deployment by applying heat to effect cure. Removing the heat source after the rigidization is complete allows the thermoset resin to cool and achieve optimum mechanical strength and stiffness. This structure may be protected from unwanted solar heating with a covering of MLI. Thermoset materials require curing temperatures similar to the softening temperatures of

thermo-plastics. However, since they require 2 to 4 hours duration at temperature their total power requirement is higher than that for thermo-plastics.

Out-gassing. Out-gassing of resin polymers is a potential concern for space formed structures. Volatile components out-gassing on orbit have the same initial velocity as the structures they out-gas from. This results in a cloud of the volatiles forming around the spacecraft. The molecules of these volatiles are in close proximity for reaction with or condensation on spacecraft surfaces. This presents a potential means for spacecraft performance degradation and has resulted in experiment out-gassing requirements. One-of-a-kind space experiments (Class D) out-gassing requirements allow a total mass loss of less than 1 percent and a 0.1 percent volatile condensable material.[7]

Out-gassing tests were performed on fiberglass test samples, at ~32% resin, using CN -104 C75, CN -120 S80 and CN - 983. The results are found in Table 1.

Table 1. Resin Out Gassing Results

Description	%Loss – 21 hr (uncured)	% Loss – 2 hr Irradiation (cured)
CN -104 C75 Sandwich	0.31	0.44
CN -104 C75 No Sandwich	1.17	1.62
CN -120 S80 Sandwich	1.82	2.31
CN -120 S80 No Sandwich	2.00	2.66
CN - 983 Sandwich	0.15	0.22
CN - 983 No Sandwich	0.10	0.18

Two sets of samples were tested: a set contained in a Mylar skin and a set without skin. The sandwiched configuration more closely resembles the flight article design. The samples were placed in a vacuum chamber at <50 millitorr pressure for a period of 21 hours. At the conclusion of this test the samples were weighed and the percent loss calculated. These samples were then replaced in the vacuum chamber at <50 millitorr pressure and irradiated for two hours with ~1 kilowatt per square meter simulated solar spectrum. This successfully cured the panels, which were then removed from the vacuum chamber and the total percent weight loss calculated. Sandwiched CN-104 C75 and both CN-983 samples met the total volatiles requirement. However, both CN – 120 S80 and the non-sandwiched CN – 104 C75 samples did not meet the total volatile requirement. Low molecular weight components such as the styrene monomers found in CN -120 S80 and the trimethylolpropane triacrylate monomer found in CN104 – C75 were found to be too volatile to meet the out-gassing requirement for space flight. Higher molecular weight polymers such as CN-983 show promise in meeting the requirements.

CONCLUSIONS

Construction of space structures using inflatable/rigidizable members is possible. Several approaches can be used for rigidization including: UV curing resins, thermoplastic, and thermoset. UV resin systems require transparent fibers such as glass or quartz. Spacecraft power required to effect rigidization of an inflated structure is greatest for thermoset materials followed by thermoplastics and much less for UV cured resins. Whereas, thermoplastic and thermoset resins are compatible with a wide variety of composite fiber compositions, such as graphite, aramid, and glass. An S glass UV resin, inflatable/rigidizable support structure for a torus-supported concentrator has been selected for the IHPRPT demonstrator due to the low spacecraft power requirement.

FUTURE WORK

Mechanical properties characterization of the S glass UV resin candidates needs to be conducted throughout the operational temperature profile. These properties will be included in a finite element model design of the space demonstration experiment for proper sizing of the support structure. A representative beam will be fabricated, deployed, cured, and tested to verify the analytical design predictions. Deployment in full sunlight, rigidization, and sun tracking of the complete inflated / rigidizable support structure are required to evaluate concept technology readiness for a space experiment demonstration.

ACKNOWLEDGMENT

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